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The USAF Electric Propulsion Program

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35th AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit
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Los Angeles, California

The USAF Electric Propulsion Program

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Abstract

An overview of the current electric propulsion technology development efforts within the United States Air Force is presented. The principal electric propulsion activity for 1999 is the successful flight of the Electric Space Experiment (ESEX), a 30 kilowatt ammonia arcjet system. This program was the culmination of a 10 year development effort to validate high power electric propulsion on-orbit and verify its compatibility with Air Force satellites. Two groups within the Air Force Research Laboratory contribute to the electric propulsion program: Propulsion Directorate and Air Force Office of Scientific Research (AFOSR). The Propulsion Directorate conducts electric propulsion efforts in basic research, engineering development, and space experiments. AFOSR funds basic research in electric propulsion throughout the country in both academia and industry.

I. Introduction

The requirement of higher satellite performance at lower cost has been a driving force towards electric propulsion, with the commercial sector so far leading the way. Within the Air Force (AF) there has been a reluctance to implement more advanced on-orbit propulsion technologies because of real and perceived risks; this is changing. MILSATCOM has baselined electrostatic propulsion for its post-2000 satellites, and Air Force SMC tested the ESEX spacecraft this year to demonstrate a 26-kW ammonia arcjet on orbit. Military satellites which could benefit from electric propulsion include: MILSATCOM, DSP, DSCS; SBIRS, a proposed Space Based Radar constellation, and a proposed orbit transfer vehicle.

Over the last several years the Air Force interest in electric propulsion was primarily directed at the north-south stationkeeping of DSCS follow-on (Advanced MILSATCOM). Chemical propulsion was baselined for all transfers to operational orbits. With the rise in available bus power in recent years, rapid progress toward maturation of high power electric thruster technology, and tolerable transfer times for combined electric-chemical orbit transfer, the use of electric technology for orbit transfer is being considered for some programs in view of the demonstrated value from a missions analysis standpoint.

II. Space Experiment - ESEX

The Electric Propulsion Space Experiment (ESEX) is a 30 kW ammonia arcjet sponsored by the USAF Research Laboratory with TRW as the prime contractor. The experiment objectives are to demonstrate the compatibility and readiness of a 30 kW class ammonia arcjet subsystem for satellite applications. ESEX is one of nine experiments on the USAF's Advanced Research and Global Observation Satellite (ARGOS). Data were acquired to characterize the thruster in four different areas: electromagnetic interactions, contamination effects, optical properties of the plume, and thruster system performance. Flight diagnostics incorporated into this experiment included: four thermo-electrically-cooled quartz crystal microbalance (TQCM) sensors, four radiometers, eight gallium-arsenide (Ga-As) solar array cells, electromagnetic interference (EMI) antennas, a video camera, and an accelerometer. ARGOS was launched on 23 Feb 99 from Vandenberg AFB, CA on a Delta II into its nominal orbit of approximately 457 nmi (846 km) at 98.7° inclination. Once on-orbit, the satellite was operated from the RDT&E Support Complex (RSC) at the USAF Space and Missile Test and Evaluation Directorate at Kirtland AFB, NM.

Preliminary results presented here indicate the system operated nominally and verified the interoperability of high power electric propulsion with nominal satellite operations.[1] Eight firings were executed mostly without incident, and the arcjet, power conditioning unit (PCU), and a propellant feed system (PFS) performed well. Ultimately, however, a anomaly occurred with the battery that precluded further firings. Since this failure occurred within days of the scheduled end of flight operations and the majority of science data had been collected, the result was only a minor impact on the overall mission success. All firings were conducted over two ground sites to facilitate ground-based observations: the 1.6m telescope at the Maui Space Surveillance Site (MSSS) for optical observations and the Camp Parks Communications Annex (CPCA) in Dublin, CA for the communication experiments.

The ESEX flight system, Figure 1, includes a PFS, power subsystem - including the PCU and the silver-zinc batteries, commanding and telemetry modules, on-board diagnostics and the arcjet assembly.

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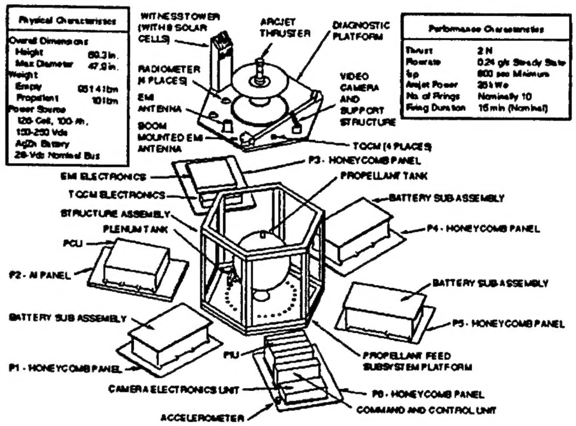

Figure 1: ESEX Exploded View

Preliminary Science Results

The science data was divided into sections corresponding to the scientific objectives and the specific sensors. These areas are optical observations, electromagnetic interactions, performance, and contamination measurements. The following data are the initial results from the experiment, and only constitute the preliminary analyses performed to date.

Optical Observations - The optical observations were made from one ground-based sensor, the 1.6 m telescope at MSSS, and one on-board sensor, the still frame video camera.[2] These sensors were used to determine the optical properties of the plume in an attempt to understand the arcjet loss mechanisms (i.e., anode heating, frozen flow losses, etc.) as well as evaluate the effects of performing similar measurements in ground-based facilities. An optical survey of the startup and ramp to full power was acquired from the on-board camera, and ground-based spectra of the arcjet firing were acquired.

The on-board camera acquired images during each of the eight firings with several different shutter speed settings. There was a significant survey of images of the arcjet during the first 90 seconds of operation which illustrate the rapid heating of the anode and the extent of the plume. These images illustrate the startup period and the majority of the 70-second ramp to full power.

The MSSS data were acquired over a series of wavelengths ranging from ultraviolet to visible. A preliminary analysis was performed on the data acquired for firing # 3 and the principal features observed from the flight agree with ground tests, namely, atomic hydrogen lines dominate the line spectrum, along with the NH complex at near-UV wavelengths; and the emission lines sit on top of a greybody continuum rising to the red end of the spectrum. Further analysis and interpretation of the results are underway.

Electromagnetic Interactions - The impacts of a 30 kW class arcjet on spacecraft communications and operations have always been a major integration concern. To address as many of these potential issues as possible, a series of tests were performed that included: measurements from the EMI antennas, communication bit error rate (BER) tests to quantify the arcjet effect on the ranging signal, and uplink/downlink tests to qualitatively verify the communication link integrity.[3] Results from the uplink/downlink test, and qualitative results from the performance of the ARGOS sub-systems, are still being evaluated, and will be presented in a future article.

The on-board EMI antennas measured the radiated emission from the arcjet in the lower gigahertz communication frequencies (e.g., S-band, X-band, etc.) The antennas sample 2, 4, 8, and 12 GHz signals with a ±5% bandpass filter on each channel. Data were gathered on the antennas for each of the firings, during quiescent spacecraft periods, and during routine spacecraft operations. The firing and non-firing data sets were then compared to identify any effects from the arcjet operation. The antenna measurements during arcjet firing periods did not differ from non-firing data corresponding with ground test data.

The bit error rate (BER) test enabled a quantified assessment of the arcjet's effect on the satellite ranging channel.[3] A series of baseline measurements were made while the arcjet was off, and with the vehicle in several transmit configurations for comparison with firing data. These error rates were all measured at transmit rates of 1.024 Mbits/sec - indicating typically less than 2 bits in 10,000 are affected.

Performance - Thruster performance was measured by three different techniques: an on-board accelerometer, ranging data from the AFSCN sites, and the ARGOS GPS receiver.[4] The performance data from each of these techniques agreed to within 1%. The Data was corrected for a suspected telemetry problem with the current shunt in the PCU.

The on-board accelerometer data were collected for all eight firings and all of the outflows. The Air Force Satellite Control Network (AFSCN) ranging data is typically used for spacecraft orbit determination in support of nominal satellite operations. For the ESEX mission, this data also determined thruster performance by comparing the orbit solutions before and after a firing. This technique provided an independent verification of thruster performance by measuring the total Δv imparted to the spacecraft.

The GPS receiver experienced difficulties on-orbit and ocassionally dropped out of navigation mode which

limited acquired data. However, preliminary GPS results agree well with the AFSCN tracking and the accelerometer data.

Contamination Measurements - An array of sensors are positioned at strategic locations of the ESEX package in order to assess the contamination effects of the arcjet firings.[5] Mass deposition, which can impact the satellite's optical and thermal control surfaces, are measured using four TQCMs. Thermal flux from the arcjet firing is measured using four radiometers coated with S13-GLO, a common thermal surface material with low solar absorptivity and high emissivity. A sample Ga-As solar array segment placed near the arcjet nozzle determines the potential for plasma-solar array interactions or obscuration of the solar flux, which can have a deleterious impact on the satellite power generation capability.

During eight firings of the ESEX arcjet, no measurable material deposition is observed that is attributable to the steady-state operation of the arcjet. The radiometers placed near the thruster exit, with a view of both the arcjet plume and body, shows material degradation of the sensor material from the arcjet firings. Solar cell segments placed near the thruster exhaust show increasing degradation through the experiment, attributable to the exhaust plasma partially shorting the solar cell load. The solar array measurements also show a 3% decrease in power generation over the time period in which the arcjet was fired, attributable to degradation in the solar transmissivity of the cover glass material. No deleterious effects associated with the arcjet firing are observed on the main ARGOS solar arrays.

In general the ESEX results are very promising for the integration of high-power electric propulsion on commercial and government satellites. Although degradation associated with contamination is observed, the effect is observed only on sensors placed near the exhaust nozzle of the thruster. It is highly unlikely that material or sensors would be located this close to the thrusters exit plane in a fully-developed high-power electric-propulsion system. Contamination sensors located in the backplane of the arcjet, or behind the thermal shield, show no deleterious effects.

Flight Anomalies

Two anomalies were observed during the flight: a liquid NH_3 ingestion observed in the PFS, and a battery anomaly that ultimately led to the conclusion of the ESEX mission.

PFS Liquid Ingestion - An ingestion of a single slug of liquid NH3 into the plenum tank occurs at the initialization of the PFS algorithm. Once the arcjet valve is opened, the plenum tank pressure and temperature soon indicate a super-heated condition and drying out of the liquid in the plenum tank as NH_3 vapor is vented. The liquid is confined to the plenum tank and never passed to the arcjet. This minor issue was easily overcome by implementing an operational delay after opening the arcjet valve - allowing all NH_3 in the plenum to vaporize before starting the arcjet. Other than this issue, the PFS performed within specification.

Injestion appears to be the result of a cold spot which allows NH_3 to condense and collect in the propellant line somewhere between the enhanced feedline heater (EFH) and the dual pressure control (DPC) valve. This phenomenon was not readily observed in ground tests and is possibly a result of a cooler mounting platform than experienced during ground tests. This platform temperature is not actively controlled, and can drift significantly - perhaps leading to a low enough temperature to condense NH_3 at the pressure in the propellant line. For an operational flight design, some heater power applied to the section of the propellant line in question would resolve the liquid ingestion.

Battery Anomaly - During the first charging cycle as the battery voltage approached a 225 Vdc potential difference, the output current from the charging circuit began cycling on and off, resulting in oscillations of the open circuit battery voltage. Beginning on firing #4, a low battery output voltage anomaly appeared which resulted in unstable PCU and arcjet operation and eventually extinguished the arc. Subsequent charge cycles showed a degrading instability that caused the charging circuit to shut off prior to attaining a full state of charge in the battery. Firing durations after #4 steadily decreased, as the battery performance deteriorated. Following the completion of firing #8, the battery voltage fluctuated erratically between 175 and 200 Vdc and ultimately the battery sub-assembly on panel #1 had a catastrophic failure.

The cause of the battery anomaly appears to be related to the mechanical properties of the interconnections between the battery cells. Preliminary results indicate the construction of the interconnections allowed the contact resistance to the cell to fluctuate and deteriorate over time (mostly as a function of temperature). This deteriorating contact resistance led to localized heating at the cell during any charge or discharge cycle, but would be greatly enhanced during the high current discharge associated with the arcjet firings. Eventually the heating ruptures the cell, causing electrolyte leakage, and the short circuit to the battery case.

ESEX Summary - The ESEX flight demonstrated that high power electric propulsion is compatible with

nominal satellite operations.[1] Although further analyses are in-work, all of the data analyzed to date indicate the thruster and the high power components operated nominally and have no significant, deleterious effect on any satellite activities. There were no negative effects observed on any of the on-board diagnostics or on the spacecraft operations.

III. Hall Thruster Programs

High Performance Hall Thruster System

The High Performance Hall System (HPHS) program is developing a 4.5 kW Hall propulsion system that provides significant payoffs for stationkeeping, repositioning, and orbit raising applications. The HPHS program is an Integrated High Payoff Rocket Propulsion Technology (IHPRPT) Phase I demonstration effort. The Air Force Research Laboratory and International Space Technology, Incorporated (ISTI) are co-funding this cost shared contract (56% government, 44% contractor). Atlantic Research Corporation (ARC) is the prime contractor. The propulsion system includes the thruster, Power Processing Unit (PPU), Propellant Management Assembly (PMA), and simulated spacecraft hardware. The Experimental Design Bureau/Fakel, a Russian designer and manufacturer of over 110 flown Hall thrusters, is developing the thruster. Space Systems/Loral (SS/Loral), leveraging their flight qualification experience with the 1.35 kW SPT-100 Hall system, is designing and manufacturing the PPU. An existing, Moog built, flight qualified PMA (not a deliverable) completes the major system components.

The development effort is focused on the improvement of thruster and PPU performance characteristics. The thruster effort has included significant magnetic system optimization. The thruster PDR and CDR were successfully completed in December 1997 and March 1999, respectively. A 1200 hour extended duration test of the flight-like Demonstration Model (DM) thruster is underway at Fakel. Thruster performance measurements at Fakel show that this thruster will meet or exceed all program objectives (55% efficiency, 7200 hour life). An extensive series of performance and contamination tests will then be conducted on the DM thruster in the United States. Performance, electromagnetic interference, and plume interaction testing will be performed at NASA Glenn Research Center under joint AFRL and NASA funding. Ion current and energy measurements will be performed at the University of Michigan Plasmadynamics and Electric Propulsion Laboratory. The Aerospace Corporation will also be performing integration testing in support of the Space and Missiles System Center's (SMC) Milsatcom System Program Office (SPO). Fabrication of the flight qualified Qualification Model (QM) thruster is expected to conclude during the 4th quarter of FY99. The QM thruster will undergo a life test, scheduled to begin during the 2nd quarter of FY00, to demonstrate a 7200 hour operating life. The PPU effort has focused on the development of improved switching topology and power magnetics design. The PPU PDR was successfully completed in November 1998, and the CDR is scheduled for the 4th quarter of FY99. Breadboard PPU test results show that this PPU will meet or exceed all program objectives (93% efficiency, 7200 hour life). The program will conclude with integrated testing of the propulsion system.

Dual-Mode Hall Thruster Program

A Hall thruster that is capable of both high Isp for stationkeeping and high thrust mode operation for maneuvering and orbit raising/topping operations is under development by the Busek Co., Inc. In a Phase I SBIR program, Busek successfully developed an 8 kW Hall effect thruster with an Isp better than 1900 sec. The Phase II effort extends the design to greater efficiency, higher Isp, bimodal operation, and longer thruster lifetime. The bimodal thruster is expected to deliver Isp's greater than 1900 sec with a thrust near 550 mN at 63 % efficiency and also be able to generate high thrust (750 mN) at an Isp of 1000 sec and 45 % efficiency. This thruster is designed for efficient operation over a broad range of Isp and thrust to satisfy the conflicting requirements for high Isp station keeping and high thrust, time critical orbit raising/topping. Bimodal operation will allow for increased payload to orbit since the high power thruster will serve dual function as primary propulsion and perform stationkeeping and fast repositioning.

COL 1 ← LOWER CASE FOR WORDS PRECEEDING AN ACRONYM

Performance measurements from 4 to 7 kW are complete and 7 to 10 kW high power testing is under way. Four different magnetic configurations are being tested. The magnetic field greatly influences efficiency and thruster lifetime and the results of the preliminary investigation will be used to design the advanced engineering design model. The thruster design process is enhanced by use of complementary analytical models. The existing 1-D model is being upgraded to Quasi 1-D by introducing radial profiles, ion temperature, and wall effects. As the program matures, an advanced engineering design model will be produced based on these preliminary studies.

Racetrack Hall Thruster

The AFOSR is currently funding Busek Co. under a Phase II SBIR contract to explore the potential of a large Hall thruster with racetrack shaped discharge cavity instead of the conventional circular cavity. A 2 kW thruster has been developed and tested, see Figure

2, which after several modifications to the basic configuration, achieved Isp =1500 sec, 38% anode efficiency at 1.5 kW and 300 V. The non-uniform features, observable in the luminosity pattern of the discharge, with the highest luminosity in the curved portion of the racetrack are not well understood presently. Attempts to increase the local B-field in the straight section, by the addition of external coils proved unsuccessful. Possible explanations may lie on electron inertial effects, forcing a higher electron temperature in the radial region and radial variations of the B-field, that are zero in the straight portion. Busek has a significant amount of design and test experience in the range of 600 W to 8 kW that gives us experimental data for comparison. At the conclusion of this program the 5kW racetrack thruster will be delivered to the AFRL for further testing.

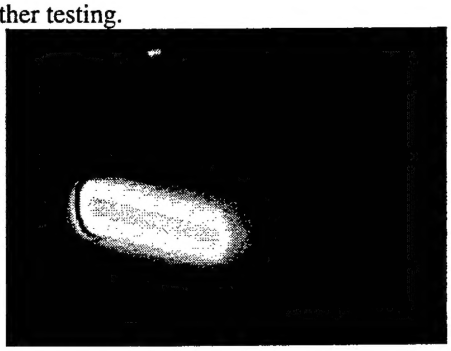

Figure 2. 2 kW Racetrack Hall Thruster firing

The racetrack has several potential advantages relative to circular thrusters including:

- linear scale up utilizing demonstrated cross section (reduced development cost)
- potential for higher performance
- easier spacecraft integration including lower beam divergence
- significant thrust vectoring without gimbals

T-40 Hall Thruster

The AFOSR is funding the development of an advanced low power Hall Effect Thruster (HET). This Phase II SBIR effort will produce, demonstrate and preflight qualify a high performance HET with an input power range from about 100 watts to 200 watts. The basic design concept is applicable to input powers of approximately 50 watts to greater than 500 watts.

Two prototype and one engineering model thrusters will be developed. The testing will include performance measurements at AFRL, plume characterization at Stanford, and erosion testing. A commercial cathode will be used for the initial development effort and an investigation of low flow cathode technology will also be performed. The effort will also include vibration testing of the engineering model thruster.

The T-40, with a 40-mm diameter discharge chamber outer diameter, is designed to produce 4 to 12 mN of thrust at ≤100 to 250 watts input at efficiencies approaching 45% and specific impulses of 1200 to 1600 seconds. The contractor's analysis shows that this thrust range and I_{sp} at low mass will also be very attractive for East-West stationkeeping on large satellites using T-160 (4.5 kW class) to T-100 (1.5 kW class) xenon Hall Thrusters for orbit transfer and North-South stationkeeping.

IV. Pulsed Plasma Thruster Programs

The PPT has reemerged as an attractive propulsion option as greater emphasis has been place on reducing satellite size for many applications. PPTs are appropriate for low power levels (<100 W) and to provide exact impulse bits for use in accurate attitude control and constellation management. The main advantage of the PPT is the engineering simplicity which leads to high reliability. This reliability has been demonstrated by the successful application of PPTs in space missions starting in the late 1960s.

MicroPPT Effort

The Micro Pulsed Plasma Thruster (Micro-PPT) is a simplified, miniaturized version of the PPT being developed at the AFRL.[6] The Micro-PPT uses a high-voltage discharge that is applied across the face of a coaxial propellant bar. Both pulsed and DC application of the high-voltage has been tested. The discharge ignites through a self-breakdown thus eliminating the PPT sparkplug and the associated mass, complexity, and energy requirements. Pulsed Micro-PPT voltage application generally tends to require lower voltage and hence reduced shielding mass. DC voltage application eliminates the mass of the semiconductor switches and voltage amplification electronics. The propellant modules consist of annular Teflon™ propellant with inner and outer copper electrodes. Module diameters of 0.110", 0.140" and 0.250" have been tested. Typical breakdown voltages for the 0.110" propellant is under 3000 V for a pulsed discharge. Thus the Micro-PPT can be energized from the trigger circuit of a standard PPT, effectively eliminating the PPU mass of the stationkeeping propulsion system. Micro-PPT discharges are typically in the 1 J regime and are estimated to create 10 μN of thrust for a 1 Hz firing rate. Thrust levels are estimated since no measurement capability presently exists at the low thrust levels required for microsatellite stationkeeping.

V. Related EP Efforts

Low Cost PPU Development

The AFRL is managing a BMDO funded effort to develop an advanced power processing unit (PPU) for

low power Hall Effect Thruster (HET) propulsion system. The Phase I SBIR effort with Space Power, Incorporated (SPI) focused on the development of a compact, low cost PPU that is capable of supporting HET power requirements for variable power operation. The effort, building upon SPI's recent development efforts funded under the NASA/BMDO funded T-160/Express program, developed the capability to operate the thruster at variable discharge voltages. The PPU was designed to support a nominal 100-200 W class HET propulsion system, including the discharge, cathode heater, cathode ignitor, thermothrottle, and flow system power requirements. The PPU will be tested at the AFRL with a laboratory model HET.

Field Emission Diamond Cathode

Busek is currently being funded under a Phase II SBIR Program to develop a diamond based field emission (FE) cathode for space propulsion applications. Motivation for field emission cathode stems from the deleterious impact of hollow cathodes on the performance (efficiency and Isp) of low power electrostatic thrusters including Hall and ion thrusters. The FE cathode is also ideal for neutralization of the colloid and FEEP thruster beams.

The Phase II effort focusses on the development of cathode emitter materials and cathode designs with sufficient current density to provide discharge and neutralization electron current for a 100 W class Hall thruster. Several materials and fabrication techniques are being investigated. Material emission requirements are influenced by total current, cathode size, and space charge limits. Contractor analysis has resulted in a baseline design for a 1 amp, 10 cm^2 emission area. The maximum electron current density is governed by the Child Langmuir space charge equation. At $V_a = 20$ volts and d = 3.5 micron, the maximum current density is approximately 17 A/cm^2. One amp in ten square centimeters is orders of magnitude less than the Child Langmuir limit.

Micropropulsion for TechSat21

There is strong interest by government agencies in reducing the size of modern satellites due to perceived advantages. This strong interest is typified by the Micropropulsion and Formation Flying Workshop sponsored by the Air Force Research Laboratory on 20-21 October 1998.[7] One potential benefit is a substantial reduction in the overall life-cycle cost by making satellites less costly to construct, due to fewer components and the potential for mass production. Smaller satellites also have greatly reduced launch costs. Employing a distributed architecture of small satellites flying in formation enables the aperture of the constellation to essentially be the diameter of the constellation (~100m) yielding much greater spatial resolution compared to a single satellite. Further, using many smaller satellites can result in graceful degradation of the system capability as individual satellites are lost and the constellation reconfigures for maximum performance.

A major new thrust within AFRL is to demonstrate the ability of a series of microsatellites (defined as <100 kg mass) to accomplish the same mission as modern, large satellites. One approach for employing microsatellites envisions them flying in formation operating in close proximity to one another in low-earth orbit. This effort is titled TechSat21 and the first application being considered is that of space-based radar (SBR). The SBR mission has been proposed for over 20 years, however the cost of deployment has been considered prohibitive. The TechSat21 approach of employing small satellites could make this long awaited DoD goal a reality. The TechSat21 program plans to launch a three-spacecraft formation in 2003, to test the critical technologies with a full-up demonstration of the space-based radar concept employing ~12 spacecraft in 2007. The critical technologies that will be validated are: 1) ionospheric effects on radar, 2) interferometric radar signal processing from multiple transmitters/receivers, 3) orbital mechanics of a formation flying constellation, and 4) spacecraft micropropulsion.

Advanced micropropulsion concepts are critical for meeting the stringent stationkeeping requirement to maintain the microsats in close proximity of each other. Individual spacecraft within the formation have slightly different orbital elements, and thus naturally respond differently to various perturbations. In order to maintain the relative positions of the spacecraft within the formation these differential perturbations, which are principally from the orbital J2 perturbation, must be corrected by periodic stationkeeping maneuvers.

TechSat21 Satellite Design - The individual spacecraft for the TechSat21 mission are currently in the conceptual design phase. The proposed design, shown in Fig. 3, collapses into a 0.3 cubic meter volume for launch, then deploys a 7-meter boom and 2.5 meter antenna on orbit. The total mass of the spacecraft is ~100 kg, of which ~10 kg is available for the propulsion system. Approximately 1 kW of electric power is produced by solar panels on the boom section, almost all of which is available for the propulsion system during the ascent and deorbit phases of the mission. FIGUR 15 M

The proposed TechSat21 spacecraft is passively stabilized by gravity gradient using the extended solar array boom, with magnetic torquers for attitude control. It is assumed that a propulsive attitude control

capability is required to conrtrol the satellite during the ascent and deorbit phases of the mission, as well as during any stationkeeping maneuvers which involve the primary propulsion system. Only a limited pitch and roll control capability will be required, primarily for damping oscillations about the stable position.

The propulsion system must accomplish an initial ascent, 10 year stationkeeping and drag makeup, and end-of-life de-orbit. Major constraints on the propulsion system are total mass, minimum impulse bit, and contamination or other interference with the constellation. The propulsion requirement are a 390 m/s total ΔV over 10-year life broken out as follows: orbit insertion ΔV of 50 m/s, drag makeup ΔV of 20 m/s, stationkeeping ΔV of 200 m/s, and deorbit ΔV of 120 m/s at end of life. Satellites are initially dropped off at 750-km altitude by the launch vehicle with on-board propulsion used to increase the orbit to 800 km and carefully insert the satellite in phase with the existing constellation. The overall system requirements to meet a 30-day insertion are, a thrust of 2 mN and a total ΔV of 360 m/s for the main propulsion system while a total of ten ACS thruster elements with a minimum thrust of 15 μN were required for the dedicated stationkeeping/ACS system, and a total ΔV of 30 m/s for the total stationkeeping/ACS system.

Micropropulsion Systems - An assessment of current micropropulsion concepts and their applicability to TechSat21 was undertaken by the AFRL propulsion directorate.[8] The study conducted a cursory look at a wide range of micropropulsion concepts, but quickly focused on those with higher specific impulse, which appears to be enabling for the TechSat21 mission. Electric propulsion systems generally offer specific impulse values of 1000-1500 seconds and chemical systems approximately 200 seconds. Miniaturized versions of conventional thrusters appear feasible for primary propulsion whereas microthrusters can meet the stationkeeping/ACS requirements.

Chemical Micropropulsion - The baseline system considered for this mission was a commercial off-the-shelf hydrazine monopropellant system that would cause minimal integration issues. Advanced monopropellant options are currently being developed by AFRL which promise to deliver up to 25% greater specific impulse than hydrazine and are considered in this study. Unfortunately, chemical propulsion suitable for use on small spacecraft are limited to approximately 325 seconds, which is achieved with MMH/NTO bi-propellant systems and results in unacceptably high propellant mass. Analysis of this option showed no mass advantages over monopropellannt systems and has the disadvantage of being more complex.

Due to the lack of throttling capability, solid rocket motors are unsuitable for stationkeeping; further, the high thrust of solids requires a correspondingly high-torque attitude control system during the ascent burn. While solid rockets offer the potential for simple, inexpensive main propulsion, the increased demands on the secondary propulsion system negate that advantage.

Cold-gas thrusters are throttleable and the simplest propulsion system suitable for the stationkeeping and attitude control requirements. Unfortunately, the combination of extremely low specific impulse and heavy, high-pressure propellant tanks results in unacceptably high total propulsion system mass.

Both TRW and Honeywell are working on the Digital MEMS (Micro ElectroMechanical Systems) thruster and presently have programs to fabricate digital MEMS thrusters and have tested the necessary igniter arrays. This device uses semiconductor manufacturing techniques to etch thousands of extremely small (~500 μm) cavities and nozzles into a silicon wafer. Each cavity is filled with a propellant charge and serves as a one-shot microthruster as needed.[9] The specific impulse and propellant mass fraction suffer in comparison with conventional chemical rockets, but the ability to scale down to arbitrarily small sizes is desirable for the microsatellite application. The availability of small discrete thrust impulses is particularly advantageous for stationkeeping and attitude control.

Electromagnetic Micropropulsion - The pulsed plasma thruster (PPT) uses an inert solid propellant, Teflon, which is attractive for microsatellites since it significantly reduces thruster mass and volume by eliminating propellant tankage and valves. The only moving part is a spring, which passively feeds the propellant., a four electrode configuration is proposed for TechSat21, all along a common thrust vector.

Significant progress has been made recently in improving PPT performance and engineering. Laboratory model PPTs have been demonstrated to achieve thrust to power levels 3 to 4 times previous flight models.[10] The next-generation of PPTs are also expected to use a coaxial geometry may better confine radiated EMI and lessen the spacecraft interaction. The radiated EMI from the PPT poses a serious concern with GHz radiation from the spark ignitor possibly interferring with primary transceiver frequencies and MHz EMI radiation from the main PPT discharge interferring with the radar frequency shifts.

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Propulsion Type			System Mass Breakdown					
Ascent	**ACS**	**Deorbit**	**Thruster**	**PPU**	**Propellant**	**Tankage**	**Misc**	**Total**
MFIT	MFIT	MFIT	0.40 kg	0.85 kg	0.25 kg	0.20 kg	0.45 kg	2.20 kg
PPT	MFIT	PPT	1.55 kg	3.25 kg	3.50 kg	N/A	1.45 kg	9.75 kg
Hall	MFIT	Hall	2.30 kg	2.30 kg	2.40 kg	1.70 kg	2.00 kg	10.70 kg
PPT	μPPT	Tether	2.95 kg	3.00 kg	3.30 kg	N/A	1.85 kg	11.10 kg
PPT	μPPT	PPT	2.50 kg	3.55 kg	4.10 kg	N/A	1.80 kg	11.90 kg
Hall	Colloid	Hall	3.00 kg	2.40 kg	2.60 kg	1.90 kg	2.20 kg	12.10 kg
Colloid	Colloid	Colloid	6.15 kg	0.85 kg	3.50 kg	0.50 kg	2.15 kg	13.20 kg
Hall	μPPT	Hall	3.70 kg	2.70 kg	3.00 kg	1.70 kg	2.45 kg	13.55 kg
PPT	MEMS	PPT	3.30 kg	3.00 kg	6.50 kg	N/A	1.90 kg	14.70 kg
Hall	MEMS	Hall	4.20 kg	2.00 kg	5.40 kg	1.70 kg	2.40 kg	15.70 kg
Hall	Hall	Hall	6.20 kg	2.15 kg	2.60 kg	1.95 kg	3.10 kg	16.00 kg
HAN	HAN	HAN	4.40 kg	N/A	12.75 kg	5.75 kg	1.95 kg	21.20 kg
FEEP	Colloid	FEEP	7.80 kg	8.60 kg	0.65 kg	0.85 kg	5.15 kg	23.05 kg
Hall	C. Gas	Hall	2.50 kg	2.00 kg	6.40 kg	10.35 kg	4.45 kg	25.70 kg
FEEP	μPPT	FEEP	9.00 kg	9.20 kg	1.45 kg	0.65 kg	5.65 kg	25.90 kg
N_2H_4	N_2H_4	N_2H_4	3.60 kg	N/A	17.55 kg	3.05 kg	2.00 kg	26.20 kg
Solid	MEMS	Solid	11.00 kg	N/A	17.60 kg	N/A	3.30 kg	31.90 kg
MEMS	MEMS	MEMS	13.20 kg	N/A	19.80 kg	N/A	3.95 kg	36.95 kg

Table 1. Propulsion System Mass Comparison for TechSat21

Electrostatic Micropropulsion - The high specific impulse, high efficiency, and modest mass have made Hall thrusters the electric propulsion system of choice for many future missions at power levels of 1 kW or above. Recent tests in the 50-200 W range suggest that they are applicable to the TechSat21 mission as well, especially for primary propulsion. However the mass associated with the accompanying power processing unit and propellant feed system make Hall thrusters marginal for stationkeeping and attitude control.

Another electrostatic thruster proposal under consideration is Field-Effect Electrostatic Propulsion (FEEP) which uses field emission ion source, in the form of a narrow slit anode through which cesium propellant is passed and ionized. The FEEP offers extremely high specific impulse values at reasonable efficiency, but specific power and thrust is low and it is necessary to relax the 30-day ascent time requirement.

Colloid thrusters consist of a series of discrete capillary tubes micromachined into a substrate. The propellant is traditionally doped glycerol and achieves specific impulse values of order 1000 seconds at reasonably high efficiency. Progress is being made in fabricating thruster subassemblies and testing representative components, although no flight hardware exists, so there would be a high degree of technical risk associated with this technology for TechSat21.

Related to the colloid thruster is the Micro Field Ionization Thruster (MFIT) from SRI, Inc. This concept also uses microvolcano field ion sources, but with a metallic propellant, typically gallium or indium. These materials melt at or near room temperature, which allows for a relatively simple feed system, while having a low ionization energy and surface tension results in the emission of single ions rather than charged clusters or droplets.[11] The current proposal calls for a specific impulse up to 15,000 seconds, with a correspondingly low thrust-to-power ratio, though it is likely that the specific impulse can be substantially reduced to meet thrust requirements. The development of MFIT thrusters is at a very early stage, and readiness for the 2003 TechSat21 mission is doubtful, but the potential benefits of such a compact, high-Isp system for follow-on missions should be noted.

Also considered for comparative purposes for the primary propulsion requirements is the electrodynamic tether which offers a substantial reduction in propulsive system mass. However, due to the immaturity of this technology and lack of understanding of the fine control for maneuvering, the tether was only considered for the specified de-orbit requirement for TechSat21.[12] Unfortunately, TechSat21 will operate in a polar orbit, whereas electrodynamic tether systems require motion perpendicular to the Earth's magnetic field to produce thrust. Although performance is substantially degraded, the tether system cac still perform the deorbit mission. There is also a strong concerns regarding tether deployment within the dynamic constellation and potential tangling with adjacent spacecraft. ← CONCERN

TechSat21 Summary - A total of seventeen propulsion options were considered, with each of the potential main propulsion systems matched with one or more compatible stationkeeping systems. Table 1 provides a comparative breakdown of all the propulsion systems analyzed in this study. The TechSat21 propulsion system proved to be mass-limited rather than power-limited. Power processing units capable of handling the full kilowatt of available power would be excessively heavy (6+ kg for the PPU alone for a 1 kW class SPT- SHOULDN'T THE TABLE

100 thruster). All of the electric propulsion systems found competitive for TechSat21 operate at power levels of only a few hundred watts.

Due to its technical maturity, high performance, ease of integration, and potential for improved performance over the next couple of years, the recommended propulsion system is the conventional Pulsed Plasma Thruster (PPT) for primary propulsion and the Micro-PPT for stationkeeping. At an estimated 11.90 kg, the combination of PPT thrusters for ascent and descent and μPPT modules for attitude control offers the lowest total system mass for any propulsion option using demonstrated technologies. Further, near-term advances in PPT technology can easily reduce the all-PPT propulsion mass to 9-kg. A low-power Hall thruster (~200 W) for primary propulsion and colloidal thruster for stationkeeping are also strong candidates.

VI. Basic Research Efforts

The AFOSR is a major funding source in the nation for basic research in electric propulsion. They are presently funding 9 universities in the areas of: PPT's, HET's, microwave thrusters and micropropulsion. An emphasis on low power has occurred within the last several years and corresponds with a national trend towards reducing the size of satellites. Summaries of many of the individual AFOSR electric propulsion programs can be found in a companion paper from this same conference.[13]

VII. Conclusions

ESEX is the culmination of over ten years of effort to validate high power electric propulsion on-orbit and verify its compatibility with standard USAF satellites. There were a total of eight firings conducted over the course of the 60-day mission, all of them over 26 kW, for a total duration of 2023 seconds. There were two anomalies associated with the flight operations - a liquid ingestion problem that had only a minor affect on the mission, and a battery failure that precluded any further firings. The battery was not a part of the demonstration aspect of this mission since an operational system would be powered directly from the spacecraft power system. Approximately 76% of the ESEX mission success was attained, with the biggest deficiencies resulting from the lack of GPS data, and a reduced number of optical signature characterization firings from MSSS. All of the demonstration aspects of the experiment were completed, and all of this hardware - the arcjet, PCU, and PFS - operated successfully, and within their specifications. All of the data analyzed to date indicate the thruster operated nominally, and operated completely independently of the normal operations of the host spacecraft (ARGOS).

On-orbit electric propulsion offers substantial benefits to both the warfighter and the taxpayer. These benefits can be realized once the risks associated with electric propulsion are both understood and minimized. Air Force missions that require electric propulsion include: precision stationkeeping for distributed satellite constellations, repositioning assets, recovering assets and re-deploying assets. The Air Force visionary report, New World Vistas strongly recommends the development of electric propulsion technologies.

AFRL continues to develop and extensive range of electric propulsion related systems including high and low power thrusters and power systems. AFRL is pushing these technologies to higher maturity levels by using advanced diagnostics, engineeering model testing, and flight demonstrations such as ESEX.

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